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TURBOMACHINERY AND COMBUSTOR TECHNOLOGY FOR SMALL ENGINES

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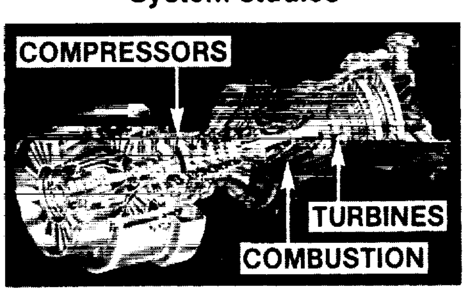
The goal of the Small Turbine Engine Technology Program is to significantly increase thermal efficiency, reliability, and durability of future small gas turbine engines. Significant fuel savings can be achieved through component and cycle improvements as well as through the use of regeneration and uncooled ceramic materials. Recent efforts to identify new regeneration concepts have not been successful, and as a result, no active regenerator research is in progress. Development of uncooled ceramic technology is taking place at the NASA Lewis Research Center under the Department of Energy funded Advanced Turbine Technology Applications Project (ATTAP). Component research includes work on the compressor, combustor, and turbine and will be emphasized in this presentation.

Significantly increase thermal efficiency, reliability, and durability of future small gas turbine engines through

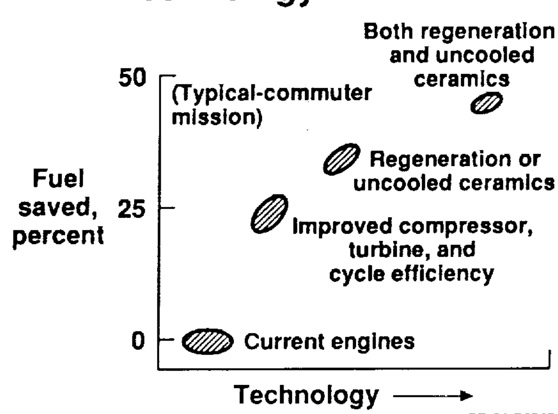
- **Improved understandings of flow physics**
- **Verification and application of advanced codes**
- **Identification and evaluation of advanced concepts for improved performance**

Research areas

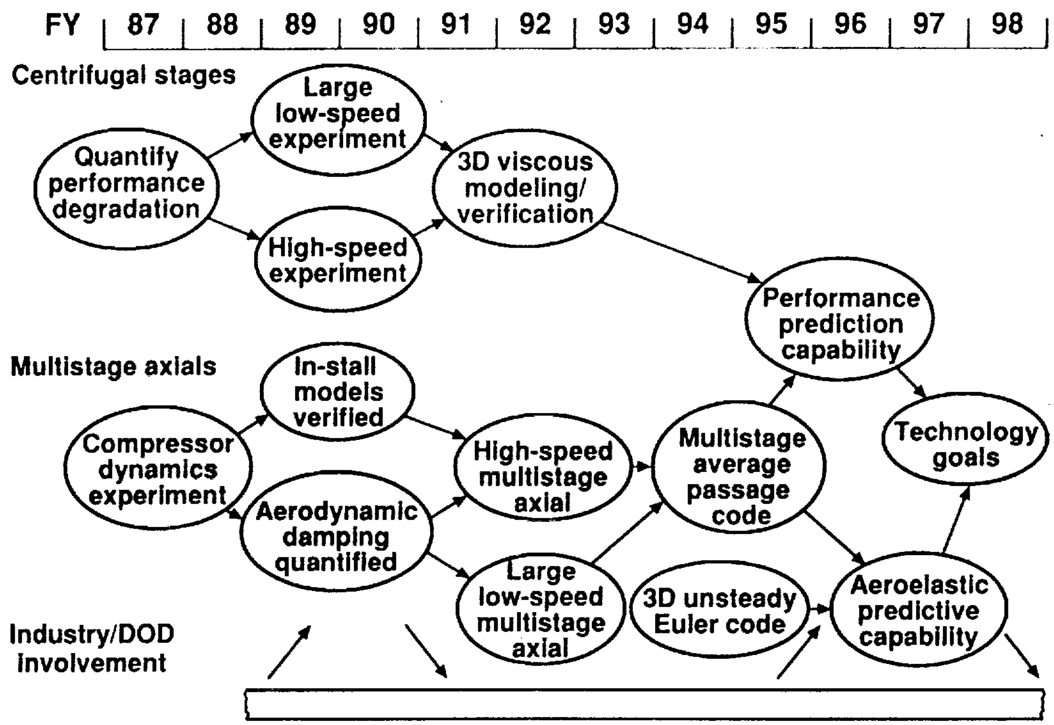
System studies



Technology benefits



Small Compressor Research Program



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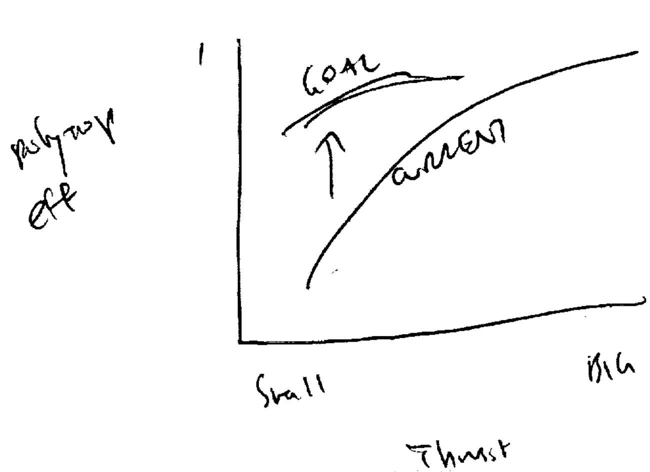
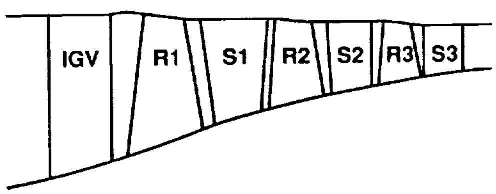
The small compressor research program is a multiyear program coordinated with industry and the military. Results from experimental programs are used to validate computational fluid dynamic codes to provide improved predictive capabilities for both centrifugal and axial compressors. The status of several elements of this program will be discussed in this presentation. The status of the large low-speed experiment will be discussed in another presentation.



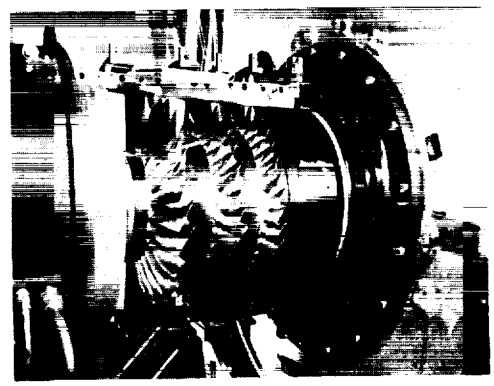
Small engines will benefit most

Compressor Dynamics Experiment

Objective
- Measure unstalled performance
- Measure stresses/determine aerodynamic and structural plus mechanical damping
- Measure post-stall performance
- Verify codes



Flowpath



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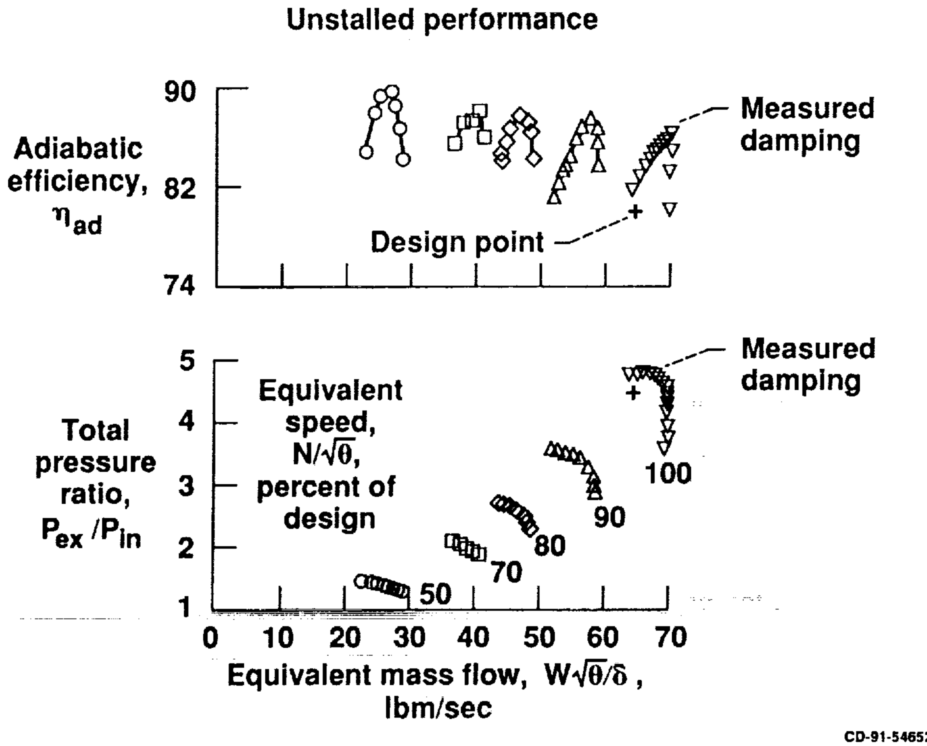
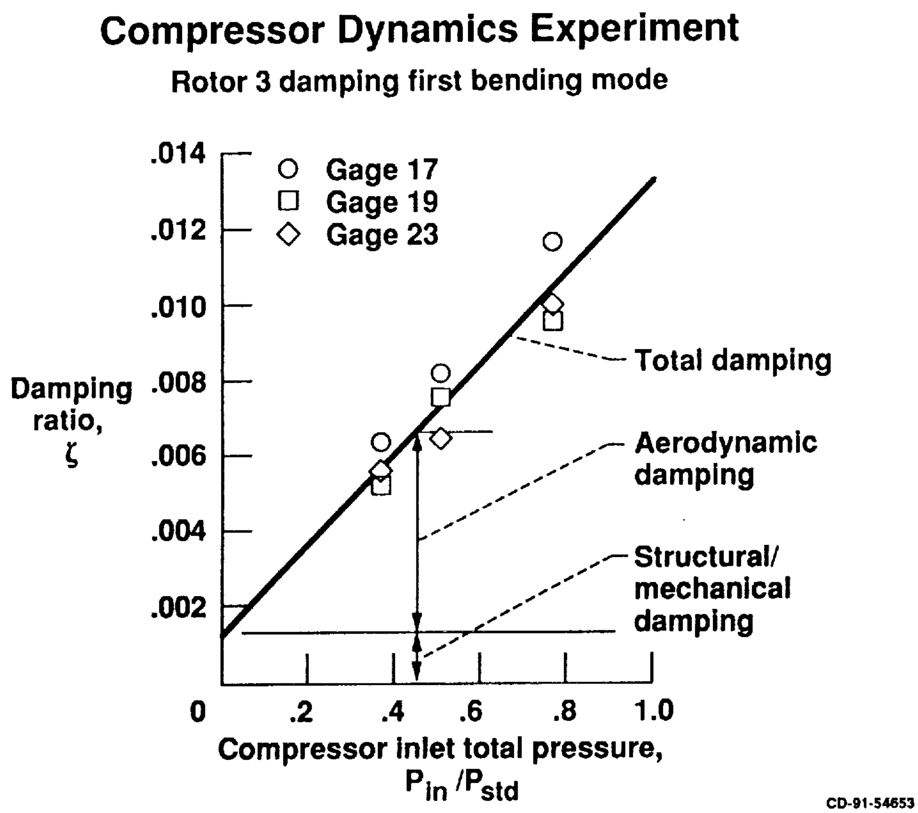
A three-stage axial compressor was tested over a range of operating conditions to determine unstalled performance, poststall performance, and aerodynamic and structural/mechanical damping of the blades. The compressor first-stage rotor has a diameter of 20 in. and a design tip speed of 1400 ft/sec. The overall pressure ratio was 4.5 at a corrected weight flow of 65.5 lb/sec.

Compressor Dynamics Experiment

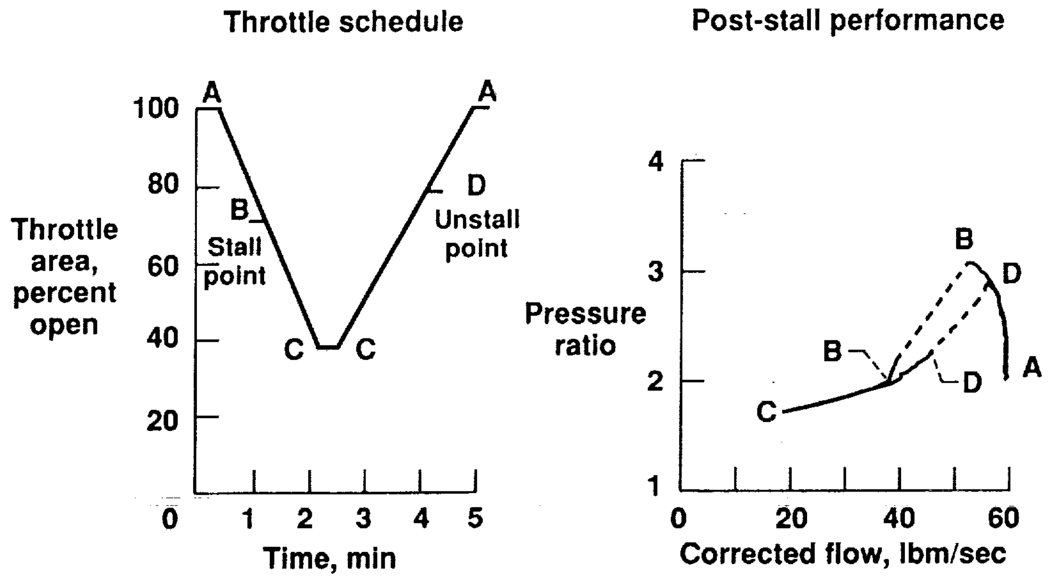


Total pressure and efficiency maps of the unstalled compressor are shown here. The design point is indicated with a "plus," and the conditions where the damping of the third-stage rotor was measured is also indicated.



The measured damping of the rotor is shown here. Damping values determined from three strain gages are shown. The damping values were determined from the strain gage output together with a one degree-of-freedom structural model. The data points and the line through them correspond to the total damping of the system, consisting of the aerodynamic damping and the structural/mechanical damping. Operating the compressor over a range of inlet pressures resulted in changing aerodynamic damping while the structural damping remained fixed. At the limit of zero inlet pressure, the aerodynamic damping goes to zero and the structural/mechanical damping is equal to the total measured damping.

Compressor Dynamics Experiment



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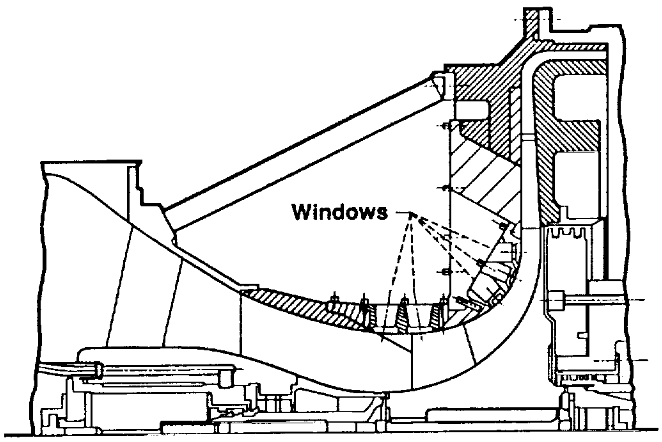
The in-stall performance of the three-stage compressor was determined during a stall induced by downstream throttling of the compressor. The throttle schedule used is shown on the left side of the figure. The throttle position was initially fully open as indicated at time zero (point A). The throttle was then gradually closed, passing through the stall (point B), and proceeding well into the stalled region (point C). The throttle was then gradually opened with the compressor unstalling (point D), finally returning to the original operating condition (point A at the right end of the curve). Compressor speed was maintained constant during this process. The resulting compressor performance in terms of pressure ratio is shown on the left side of the figure. At compressor stall there is a sudden drop in pressure ratio and mass flow (from the upper point A to the lower point A), and at compressor unstall there is a sudden rise in pressure ratio and mass flow (from the lower point D to the upper point D). This data is being used to evaluate poststall performance prediction models.

4:1 Pressure Ratio Compressor

Objective:
Obtain aerodynamic performance and laser anemometer measurements of the flow fields within a high-speed centrifugal compressor.



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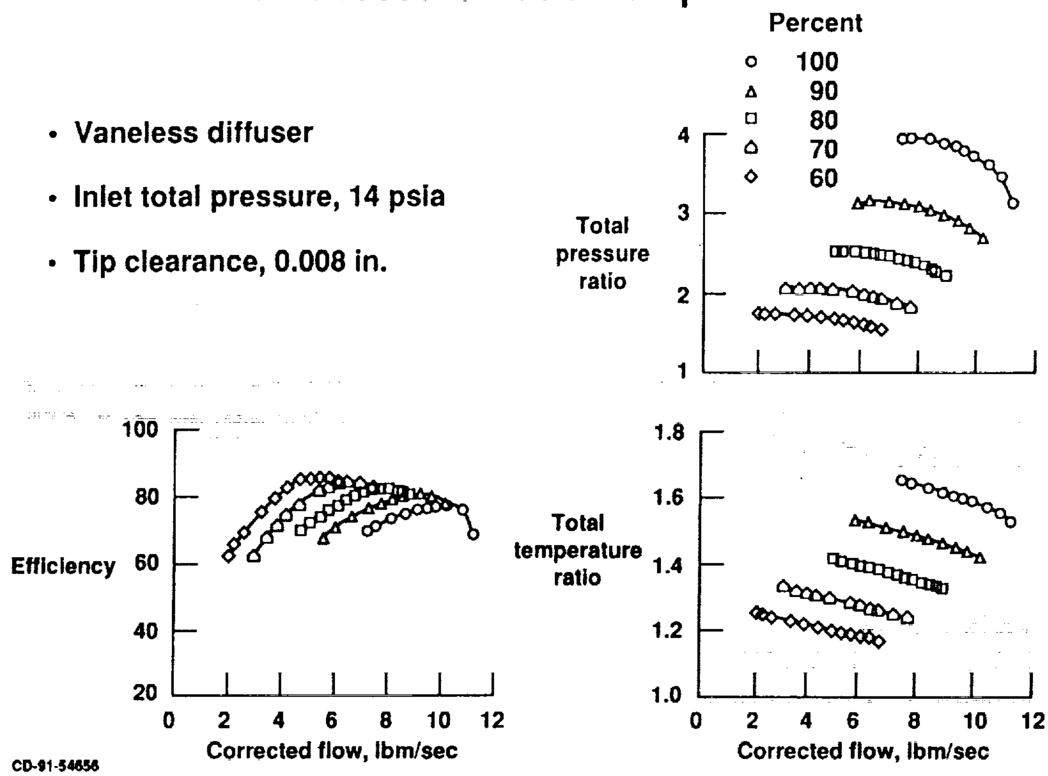
A small centrifugal compressor is being tested to provide data that will allow an improved understanding of the flow fields in centrifugal compressors. The compressor is a scaled-up version of an Allison design. At the design point corrected tip speed of 1600 ft/sec and corrected weight flow of 10 lb/sec, the compressor had a pressure ratio of 4:1. This compressor incorporates splitters in the rotor blade passages. In addition to measuring overall performance, detailed flow-field measurements will be made in the rotor blade passages. A photograph of the compressor and a cutaway drawing of the hardware are shown here. Areas indicated by arrows are casing windows which allow optical access to the rotor blade passages for laser velocimetry measurements.

4:1 Pressure Ratio Compressor

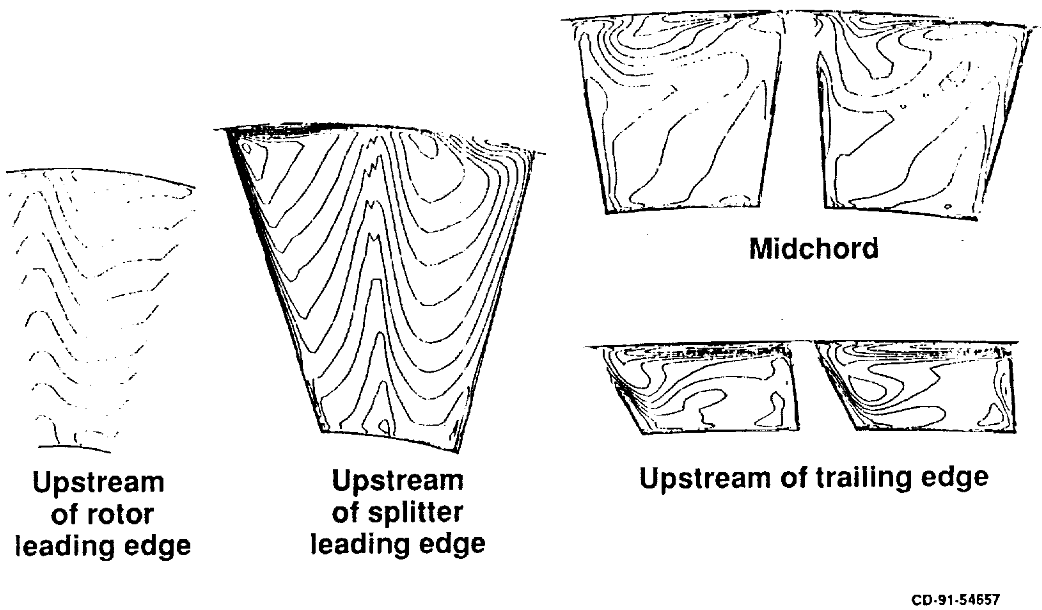
- **Vaneless diffuser**
- **Inlet total pressure, 14 psia**
- **Tip clearance, 0.008 in.**



At this time, the compressor has been tested with a vaneless diffuser. Experimental results for this configuration at atmospheric inlet pressure and design tip clearance are shown in the figure. Pressure ratio, temperature ratio, and efficiency maps are included. Tests have been conducted at several inlet pressures from 3 psia to ambient and also with several spacings between rotor and casing ranging from design to 5 times the design value. The compressor will be tested next with a vaned diffuser followed by laser velocimetry measurements in the rotor blade passages. There are currently no detailed velocity measurements for a centrifugal compressor with splitters. Thus, it is very important to obtain laser velocimetry data to allow validation of the three-dimensional, viscous CFD codes that are being used to predict the flow in this type of compressor.

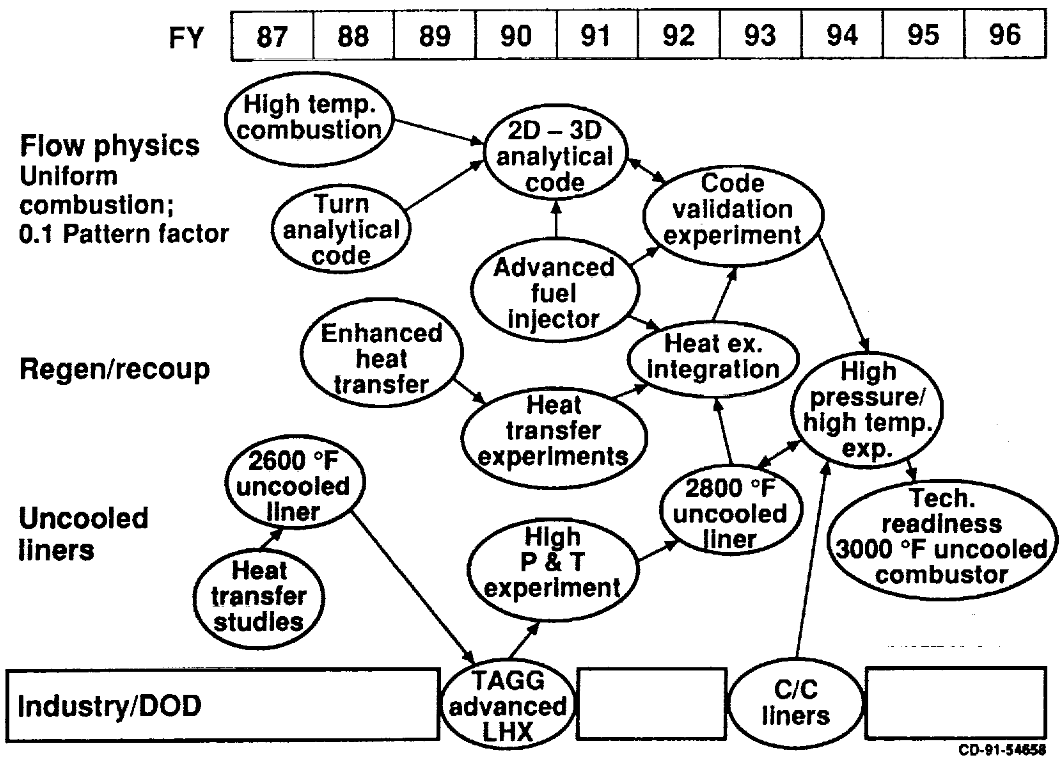
4:1 Pressure Ratio Compressor

Predicted Mach Number Contours



Three-dimensional, viscous calculations of the flow field in the 4:1 pressure ratio compressor have been performed using the turbomachinery code developed by Dawes. Tip clearance for both the blade and splitter are included in the calculations. Flow-field predictions are shown in this figure for a flow rate near the design value. Mach number contours are shown at several planes approximately perpendicular to the flow path. Results are shown slightly upstream of the rotor blade leading edge, just upstream of the splitter leading edge, about midway between the splitter leading and trailing edges, and slightly upstream of the blade and splitter trailing edges. For each contour plot, the hub surface is at the bottom, the shroud surface is at the top, the blade suction surface is at the left, and the blade pressure surface is at the right. The splitter is in the center of the passage. The results indicate that the flow upstream of the splitter exhibits the expected higher velocities near the blade suction surface compared to the pressure surface and a slightly reduced velocity at midpassage ahead of the splitter leading edge. Just ahead of the blade and splitter trailing edges, the gradients present ahead of the splitter leading edge have disappeared, and the flow in both passages is similar with only minor differences present.

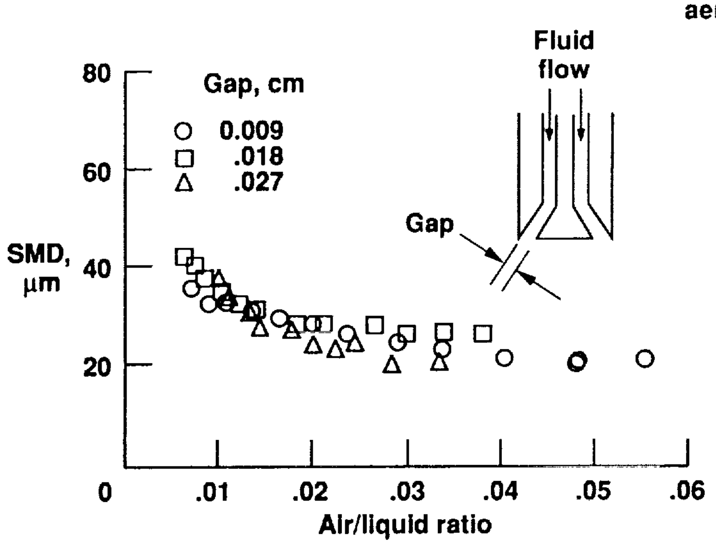
Reacting Flow Research

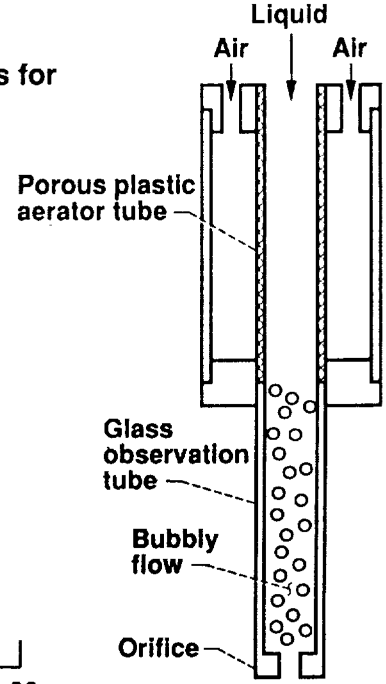


The overall program in combustor research for small turbine engines is summarized in this figure. Research on the flow physics of reacting flows emphasizes analytical and experimental studies aimed at providing very uniform flow fields at the combustor exit. Research on uncooled liner technology is in progress to allow operation at 3000 °F. Research efforts are coordinated with industry and the military. Combustion research on regenerators and recuperators has been delayed because efforts to identify attractive new concepts for these components have not been successful. However, new ideas, which would allow more efficient or lighter weight regenerators and recuperators, are still being sought. Research in progress in several areas will be summarized in this presentation.

Effervescent Fuel Injection

Objective:
Evolve advanced fuel atomization techniques for small gas turbine combustors.





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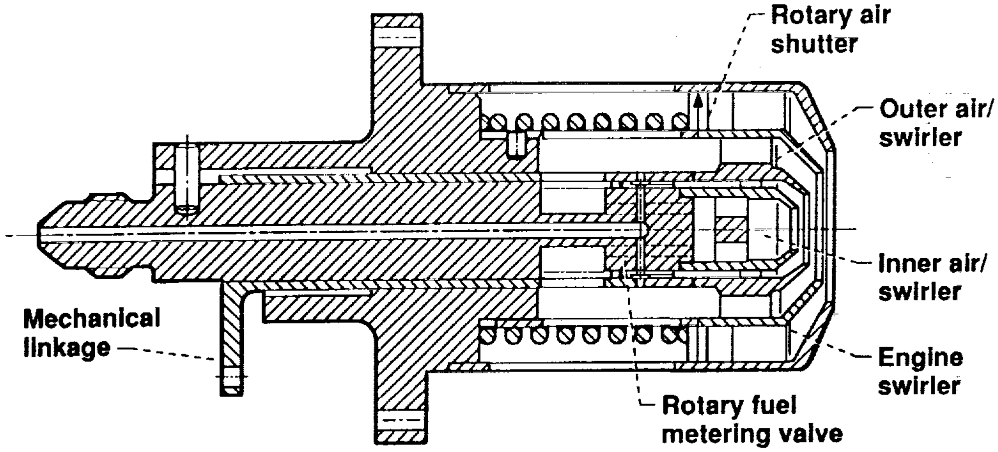
An experimental program is in progress to develop advanced fuel injectors with improved fuel atomization characteristics. The approach shown conceptually at the right involves mixing air with the fuel prior to injection in the combustor. This mixing is accomplished by using a porous aerator tube. Results for an experimental injector tip configuration shown at the left indicate that small droplet sizes can be obtained for a wide range of injected air to liquid fuel ratio and that the droplet size is not sensitive to the size of the injector flow passages. In addition, small droplet sizes are available with fairly large passages, an advantage for small engines.

Advanced Fuel Injector

Objective:
Evolve advanced fuel injection technology for future small gas turbine applications capable of doubling the operating range of current fuel injectors.



Variable geometry pure airblast

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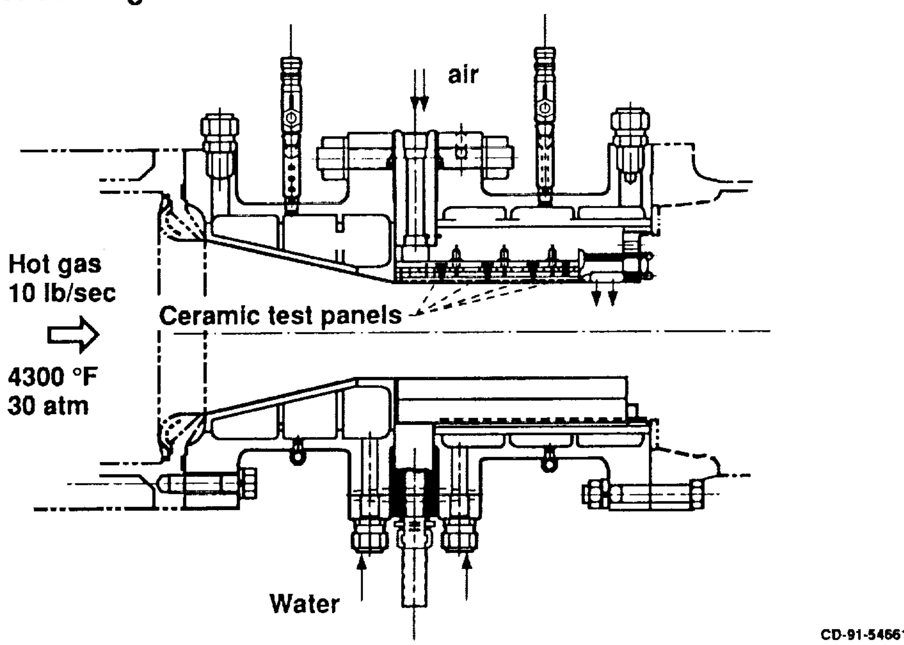
Future aircraft missions will require combustors to provide considerably higher temperature rises. This requirement imposes severe problems for fuel injectors that will be required to operate over wider operating ranges. A contractual program was formulated to meet this need by evolving injectors that double the operating range. An example of one of the injectors being developed in this joint NASA/Army/Allison program is shown in this figure. The injector is an air blast type. Kinetic energy of the combustor airflow is used to break up the fuel into small, uniform droplets conducive to uniform combustion. Air blast injectors are low fuel pressure loss devices in contrast to conventional pressure atomizers that are high fuel pressure drop devices. One of the major advantages of air blast atomizers is that small restrictions, prone to clogging, are eliminated.

Ceramic Matrix Liner Test Rig

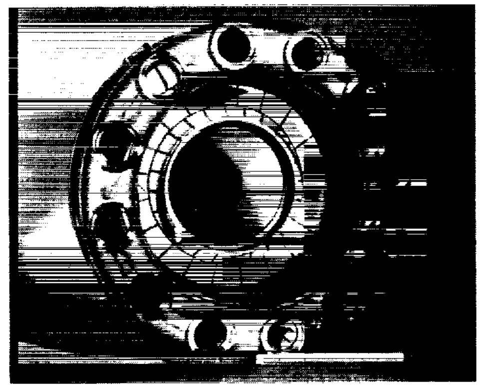
Objective:
Evaluate advanced ceramic/composite materials under advanced engine conditions.

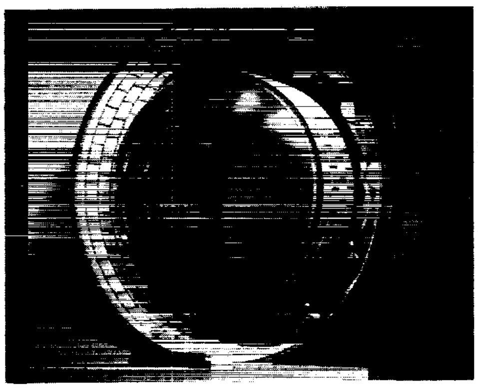


A new test rig for evaluation of ceramic/composite combustor liner materials under realistic engine conditions has been developed at the NASA Lewis Research Center. The test rig can evaluate up to 16 samples simultaneously. The samples are placed in four quadrants, each of which has independently controlled coolant flow rates. The samples are exposed to a hot gas source at high pressure to simulate combustor operating conditions. This approach allows the controlled testing of materials, apart from the issue of combustor design and characteristics, such as gradients in combustion gas temperature and composition. This test rig facility provides an excellent opportunity for joint programs between NASA and engine or ceramic/composite manufacturers for evaluating new combustor materials.

Ceramic Matrix Combustor

- Advanced cycle 3000 °F exit temperature
- Compact reverse flow annular design
- State-of-the-art design methodology
  - 3D internal reacting aerodynamics
  - 2D wall temperature distribution





Payoffs
- 50% reduction in cooling air requirement compared to Lamilloy or effusion
- Low emissions and smoke
- Improved efficiency, pattern factor, and durability

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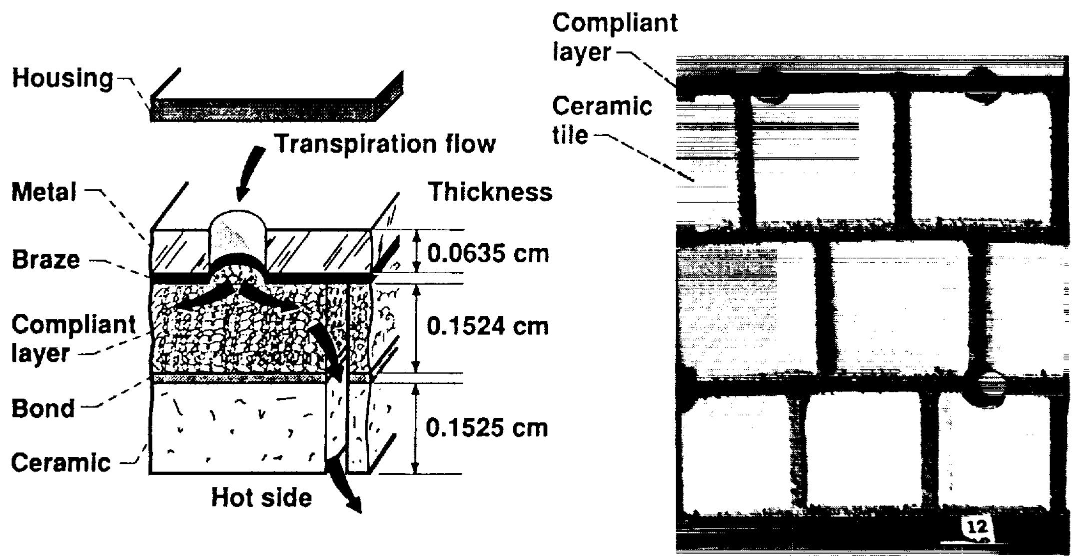
An advanced liner concept, the ceramic matrix liner, was applied to a combustor as shown in the figure. The combustor is a compact, reverse flow, annular combustor designed with a three-dimensional reacting flow computer program and a two-dimensional heat transfer analysis. This advanced ceramic matrix combustor has the potential of reducing combustor cooling air by 50 percent compared to Lamilloy or effusion cooled combustors. In addition, the combustor promises low emissions and smoke as well as improved efficiency, pattern factor, and durability. The ceramic material is visible in the photographs as small light colored rectangular features on the interior surfaces of the combustor. This research, performed under a joint NASA/Army/Allison program, has applicability to future DOD engines.

Ceramic Matrix Combustor



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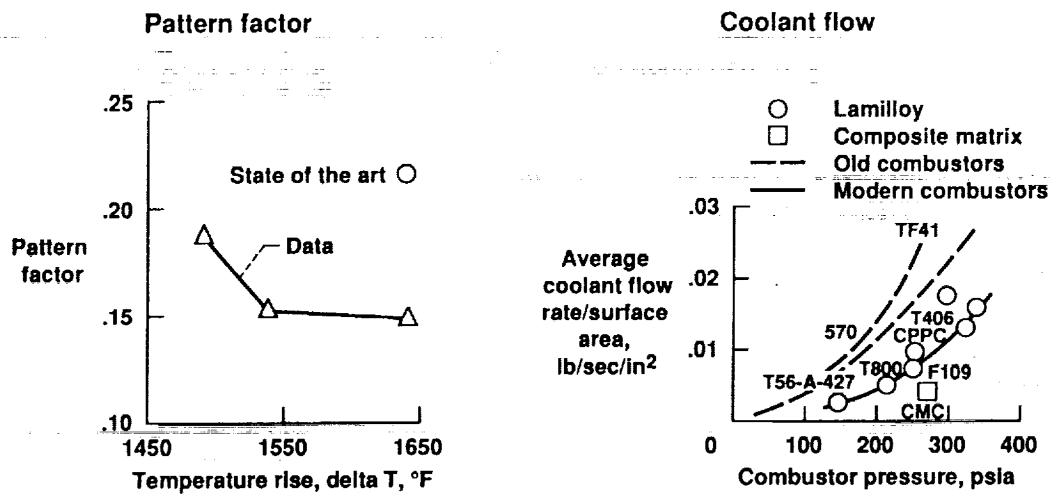
A cross section of the ceramic matrix combustor liner is shown here on the left. Structural loads in the combustor are carried in a metallic substrate. The ceramic liner material is attached to this metallic substrate with a porous, compliant pad that allows cooling flow to pass between the ceramic tiles and isolates the ceramic tiles from structural loads. The cooling air enters through holes in the metallic substrate, passes through the porous pads, and exits between the individual ceramic tiles into the combustor. On the right, the figure shows a photo of a portion of the combustor interior with the ceramic tiles and coolant passages. The tiles are attached to the compliant layer which can be seen through the passages.

Ceramic Matrix Combustor
Test results



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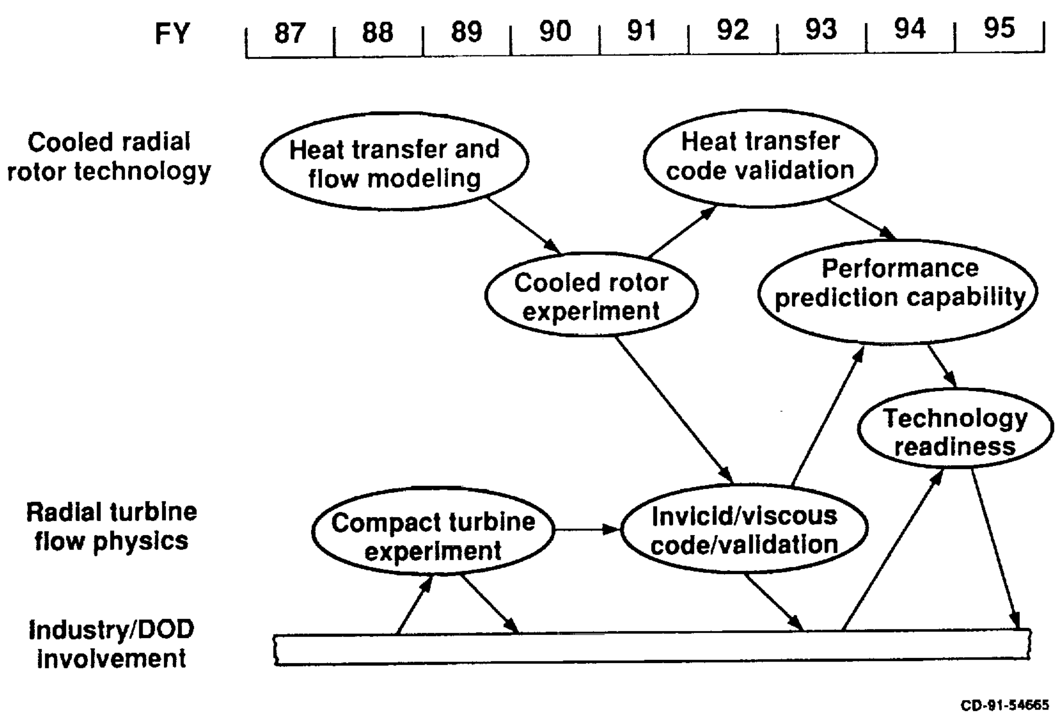
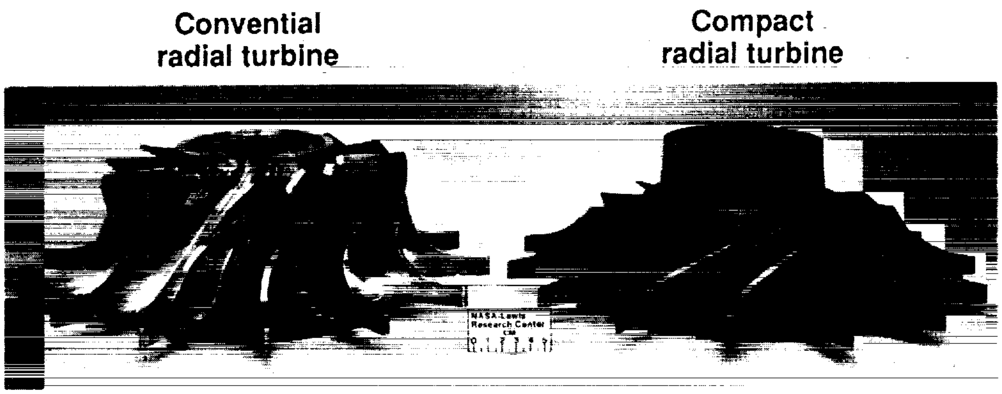
Test results shown on the right of the figure demonstrate that approximately 50 percent lower coolant flow rates are required for the ceramic matrix combustor liner compared to a modern Lamilloy combustor. Even larger reductions relative to older design combustors are possible. Since less liner cooling flow is required, more airflow is available for other combustor processes. Introducing this "saved" air into the combustion zone results in more uniform burning that produces the reductions in pattern factor shown on the left side of the figure.

Small Turbine Research Program



The small turbine research program is a multiyear program coordinated with industry and the military. Results from experimental programs are used to validate computational fluid dynamic and heat transfer codes to provide improved predictive capabilities for radial inflow turbines. The status of several elements of this program will be discussed in this presentation. Although the turbine research program has emphasized radial inflow turbines for the last several years, the program is being expanded to include work on axial flow turbines. A transonic linear turbine cascade facility will be described in another session. Plans for turbine testing using rotating rigs are in the formulation process.

Conventional/Compact Radial Turbines



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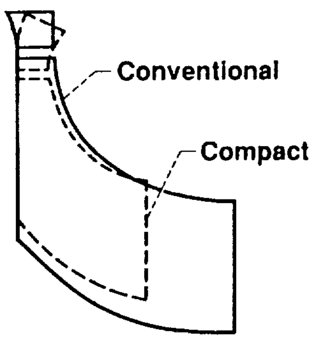
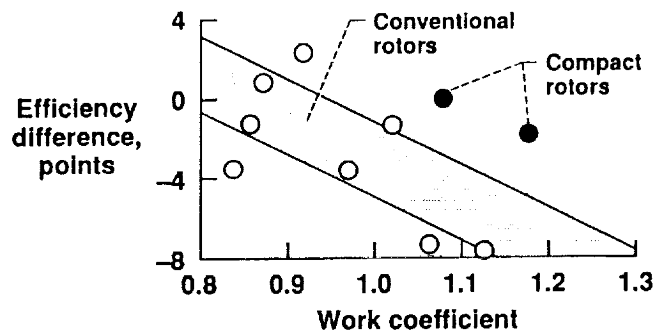
A compact radial turbine, 40 percent shorter than conventional radial inflow turbines, was designed and fabricated by Pratt Whitney and tested at the Lewis Research Center. A conventional radial turbine is shown here on the left and the compact radial turbine is shown on the right. The compact radial turbine was built with a 14.4-in. tip diameter and has a design point pressure ratio of 5:1. The reduced length of the compact turbine offers significant weight savings relative to the conventional turbine.

Compact Radial Turbine

Objective:
Conduct fundamental experiments to obtain detailed data for model development and code validation of compact/lightweight radial turbine.





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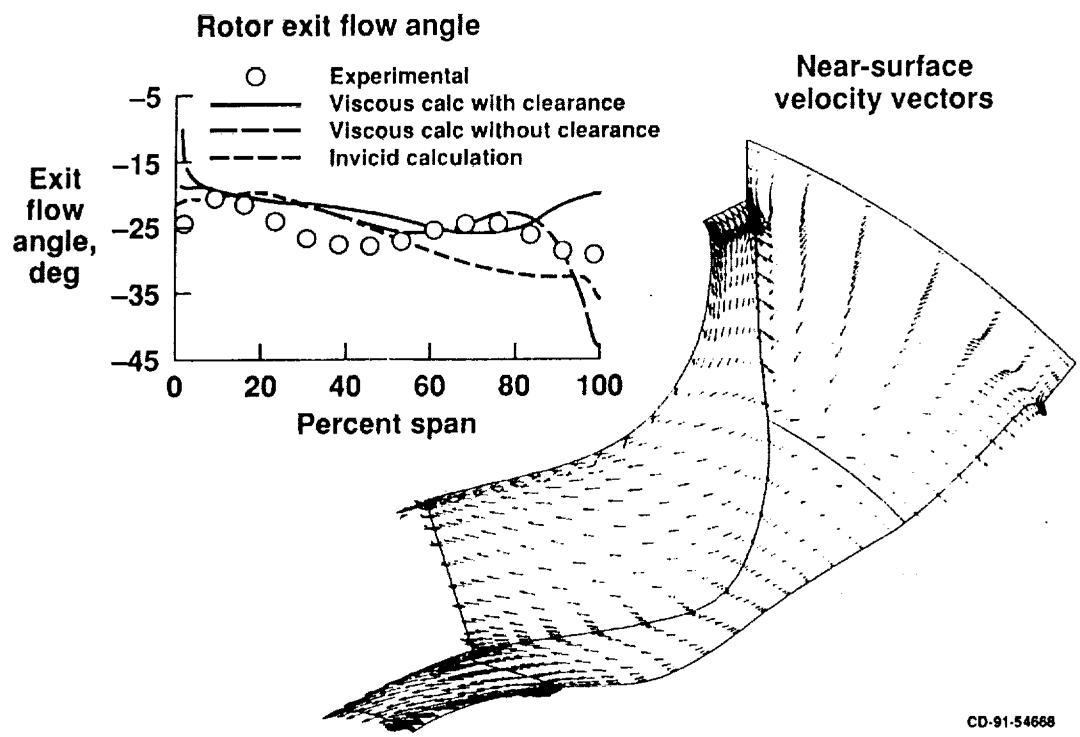
Another comparison of the compact and a conventional radial inflow turbines is shown in this figure. Test results for the compact turbine, shown here as symbols, indicate that the efficiency is comparable to, or somewhat higher than, that for a conventional radial inflow turbine over a range of operating conditions. The measured efficiency levels, together with the short length and reduced weight of the compact radial turbine, make this concept a very attractive one for future radial inflow turbine applications.

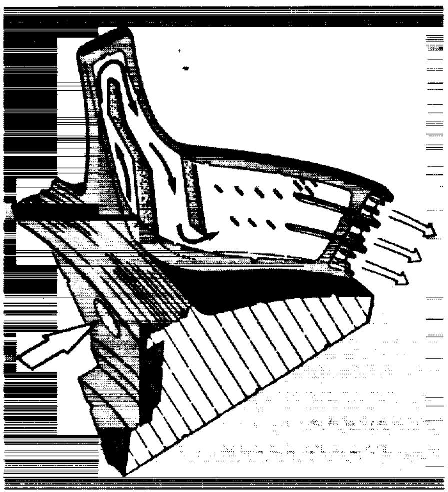
Compact Radial Turbine Analysis



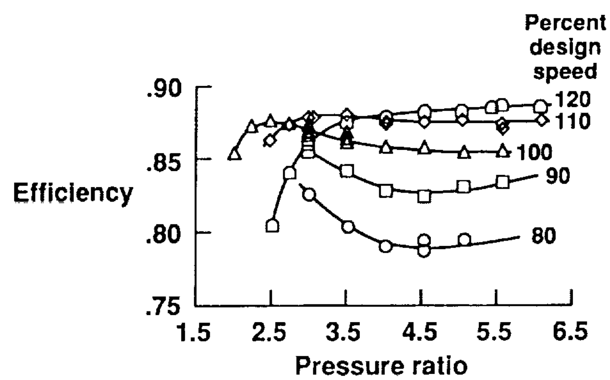
The compact radial turbine has been analyzed using a three-dimensional average passage analysis developed at Lewis by Adamczyk. This code, which was previously used only for axial turbomachinery components, was modified to allow radial inflow geometries and blade tip clearance. Computational results were obtained for various levels of modeling sophistication including inviscid with no clearance, viscous with no clearance, and viscous with clearance. A portion of the calculated results are shown on the left side of the figure where turbine exit flow angle is plotted versus percent span for these calculations. Results indicate the sensitivity of the calculation to viscous effects and blade clearance. Although the viscous flow results with tip clearance are in reasonable agreement with the experimental data, it should be noted that the clearance used in the calculation was not an exact match to the actual tip clearance distribution present in the experiment. Based on the sensitivity of the results to including clearance, it is reasonable to assume that an accurate specification of clearance distribution will be required to accurately predict turbine exit flow-field characteristics. Shown on the right side of the figure are velocity vectors near the rotor and hub surfaces from the viscous calculation with blade clearance. The velocity vectors near the intersection of two surfaces show the effect of the tip clearance flow as a sudden change in flow direction.

Cooled Radial Turbine

Objectives
- Develop rotor cooling technology to increase operating temperature of radial turbines
- Demonstrate the fabrication of a cooled radial turbine
- Obtain data to assess and improve heat transfer and aerodynamic codes



Stage performance of uncooled configuration of cooled radial turbine



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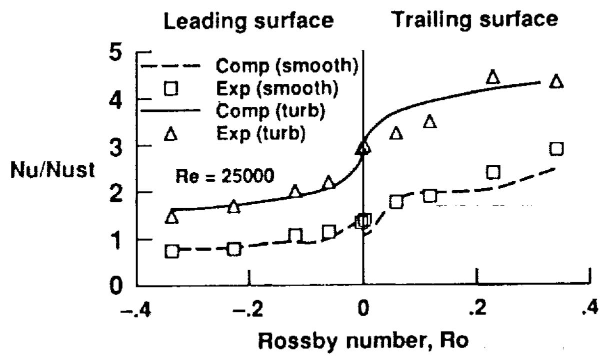
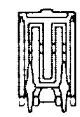
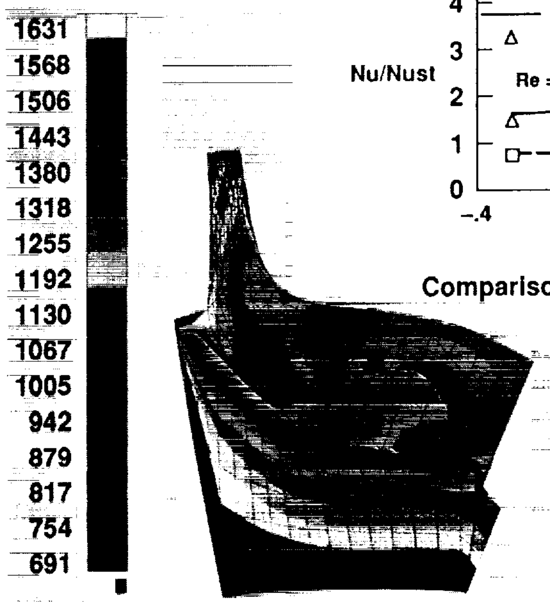
Although cooled blades for axial flow turbines are quite common, radial inflow turbines do not currently use cooled blades. The complex fabrication techniques required to build a cooled radial inflow rotor are being addressed by the NASA Lewis Research Center under a contract to the Allison Gas Turbine Division of General Motors. The use of a cooled radial turbine will allow the use of higher turbine inlet temperatures in small engines. Under this program a turbine rotor was fabricated with integrally cast cooling passages. Final machining of the rotor is in progress. In the meantime, a solid rotor was tested to provide aerodynamic performance data. The turbine, built with a 14.4-in. tip diameter, has a design point pressure ratio of 4:1. Turbine efficiency results are shown here. The performance is typical of a modern design radial turbine. The cooled version of the turbine will be tested in a warm turbine test rig in which the ratio of coolant temperature to turbine inlet temperature will be the same as in an actual engine. In addition to measuring the performance of the cooled radial inflow turbine, the program objectives include obtaining detailed data for validation of design and analysis codes for aerodynamics and heat transfer. Additional tests are planned for acquiring rotor surface static pressure information, flow-field data using laser velocimetry, and coolant passage heat transfer data.

Turbine Coolant Passage Heat Transfer Analyses





Comparison with rotating serpentine passage experimental data



Thermal analysis of cooled radial turbine

Analysis features

- Rotational effects
- Entrance and turning effects
- Leading edge impingement
- Ribbed turbulators
- Pin fins (including lateral ejection)
- Flow branching
- Finned passages

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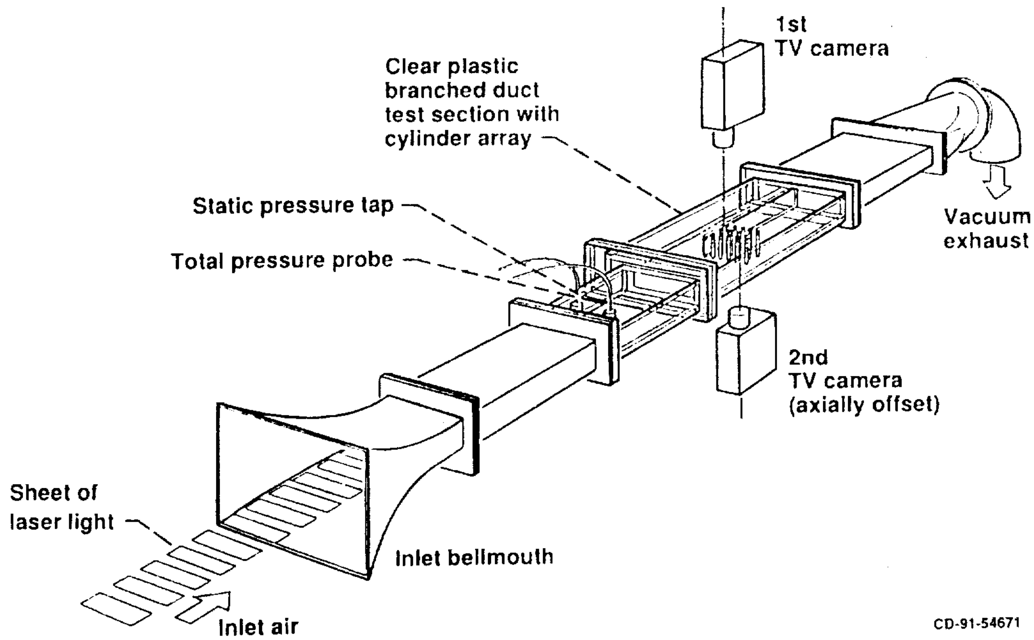
To better understand the aerodynamics and heat transfer in turbine blade cooling passages, a one-dimensional analysis for aerodynamics and heat transfer has been developed. This analysis includes many physical and geometric features that are used for turbine blade cooling passages. A list of these features is included on the figure. The analysis has been applied to a rotating serpentine cooling passage for which heat transfer data are available. Analysis results are compared to the experimental data. Excellent agreement between experiment and analysis was obtained for both smooth and rough cooling surfaces. This coolant code was used in making a rotor thermal analysis of the cooled radial inflow rotor described in the previous figure. The predicted temperature distribution on the surface of the blade is shown. Heat transfer results from the cooled turbine test program will be used to further validate the analysis.

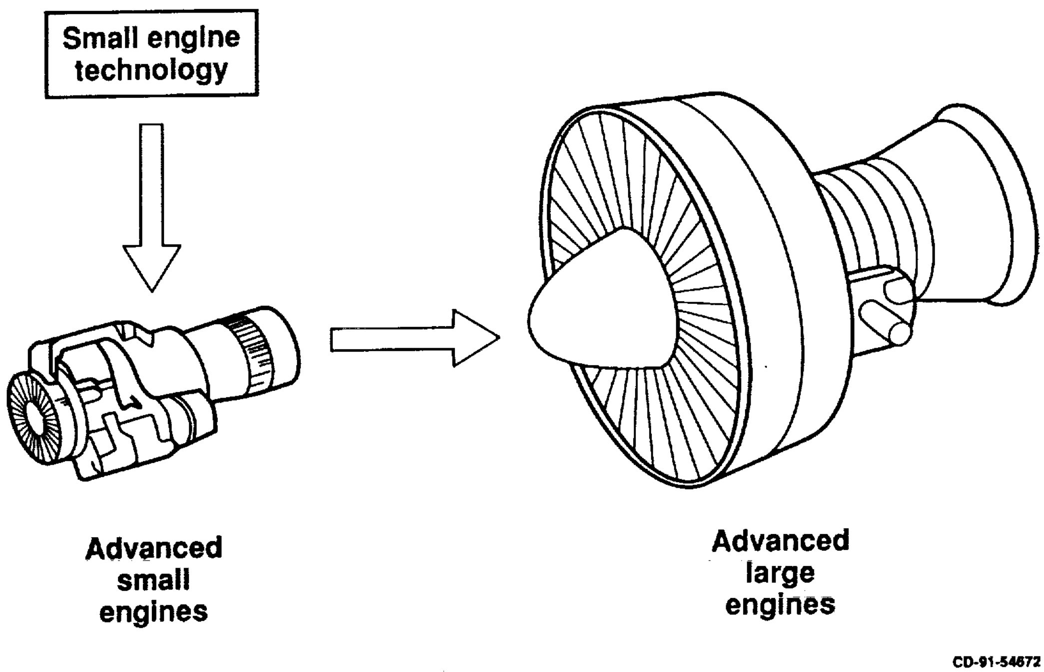
Turbine Cooling Passage Experiment

Objective:
Obtain detailed flow-field and heat transfer measurements in a simulated turbine cooling passage.



As coolant passage analyses become more sophisticated, more detailed flow-field and heat transfer measurements will be required to validate the analyses. To provide this kind of detailed data, a relatively large simulated cooling passage is being built to allow measurements in a simplified, but representative, geometry. The experimental apparatus is shown in the figure. The flow passage divides and one branch of the passage contains pins similar to those which would be used in a radial inflow turbine to increase turbulence and heat transfer. In addition to detailed surface static pressure and heat transfer measurements, flow visualization will be performed using laser light sheets.

Small Engine Technology and Large Engines



The immediate beneficiary of the NASA small engine program is the small engine manufacturer. However, as manufacturers of engines for large commercial and military aircraft strive for higher efficiency through higher overall pressure ratios, the high pressure compressor blades, the combustor, and the high pressure turbine blades can become quite small. These components can approach the size of those used in the small engines of commuter and general aviation aircraft. Because of this, the NASA small engine program, in addition to advancing technology for small engines, is laying the groundwork for future advances in large engine technology.

CONCLUDING REMARKS

Turbomachinery and combustor research in progress at the NASA Lewis Research Center for small engines has been presented. Steady-state performance for both stalled and unstalled operating conditions were measured for a three-stage axial compressor. In this same experiment aerodynamic and structural/mechanical damping were measured. Performance measurements are in progress for a single-stage centrifugal compressor, and detailed flow-field measurements are planned to provide a better understanding of centrifugal compressor flow fields. Advanced fuel injector designs are being investigated experimentally to provide improved fuel atomization and increased operating range. Ceramic and ceramic composite materials are being investigated in coupon testing and in an actual combustor to allow increased combustor temperature or decreased combustor cooling flow. An advanced, compact radial inflow turbine was tested to demonstrate the ability to reduce turbine length while maintaining performance. An experimental program to demonstrate the feasibility of a cooled radial turbine is in progress. A simulated cooling passage experiment as well as a warm turbine test are planned. State-of-the-art computational fluid mechanics and heat transfer methods are being used in conjunction with the experimental programs to gain a better understanding of the physical processes involved in small engine components. In addition, improved analysis methods will result, which will allow the design of improved components.